FIRST QUARTERLY PROGRESS REPORT

JPL Contract No. 951574

POWER SYSTEM CONFIGURATION STUDY AND RELIABILITY ANALYSIS

Period Covered:

7 July 1966 through 7 October 1966

This work was performed for the Jet Propulsion Laboratory, California Institute of Technology, sponsored by the National Aeronautics and Space Administration under Contract NAS 7-100.

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by

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ABSTRACT

Model spacecraft configurations and sets of power requirements were determined for five interplanetary missions, namely, 0.3 AU and 5.2 AU probes, and Venus, Mars and Jupiter orbiters. Mission profiles and power profiles were defined for each configuration. Representative solar array current-voltage output characteristics were calculated for each mission. Analyses of optimized photovoltaic power system configurations based on maximum reliability and minimum weight were initiated. Candidate baseline (non-redundant) power system configurations were determined for each model spacecraft. These configurations are characterized by different regulation and control techniques to integrate the solar array and battery, and supply regulated outputs. The various investigations are summarized and examples of the results are provided.

DISCLAIMER CLAUSE

NOTICE

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1. INTRODUCTION

This is the first quarterly progress report covering work performed by TRW Systems under JPL Contract 951574, "Power System Configuration Study and Reliability Analysis." This report summarizes the study effort during the period 7 July 1966 through 7 October 1966.

The principal objective of this study project is the development of photovoltaic electric power system design optimization data and procedures for five interplanetary missions: 0.3 AU and 5.2 AU probes, and Venus, Mars, and Jupiter orbiters. The project is divided into the following tasks:

Task I: Model Spacecraft Requirements

(a) Mission Analysis. Analyze the five specified missions to determine spacecraft configurations for each, based on booster capabilities, mission objectives, and subsystem requirements.

(b) Power Requirements. Analyze model spacecraft configurations to establish load power requirements including power profiles and characteristic voltage levels and regulation limits.

Task II: Baseline Power System Configurations

(a) Solar Array Analysis. Determine current-voltage characteristics of solar array as functions of mission time for each model spacecraft.

(b) Analysis of Baseline Systems. Define alternative baseline (nonredundant) power system configurations which are compatible with each of the spacecraft models. Determine the principal advantages and disadvantages of each with respect to reliability, weight, spacecraft integration, efficiency, complexity, and flexibility.

Task III: Power Systems of Improved Reliability

(a) Methods of Reliability Improvement. Perform component and system failure mode analyses for each baseline configuration and establish methods of improving component reliability.

(b) Effects of Reliability Improvement. Investigate and describe effects of reliability improvements on component reliability, weight and efficiency, and system weight and reliability.

Task IV: System Recommendations

Compare alternative system configurations from Task III to select those providing maximum reliability as a function of weight. Recommend an optimum configuration for each model spacecraft.

Task V: Telemetry Criteria

Investigate telemetry monitoring points, parameter ranges, and priorities for various system configurations from Task III. Investigate utilization of telemetry data during both normal and abnormal system operation. Develop generalized criteria for power system telemetry requirements.

In addition to a final report which will fully document all study efforts, a "Spacecraft Power System Configuration Reference Manual" will be prepared to provide a design reference for use in the determination of optimum power system configurations for various interplanetary missions.

2. PRESENT STATUS OF THE STUDY

The study efforts completed during the first quarter represent approximately 20 percent of the total planned engineering effort. Task I, the determination of model requirements, is complete. Task II, the analysis of baseline power systems for each model requirement, is approximately 30 percent complete.

The project schedule is shown in Figure 1.

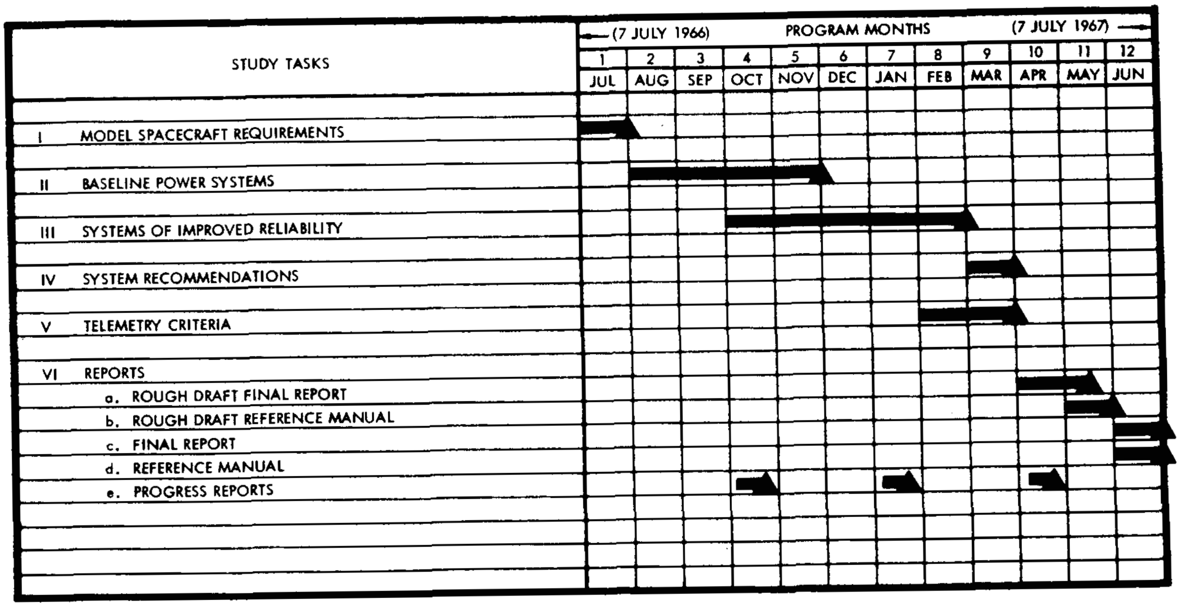

Figure 1. PSC Study and Reliability Analysis Project Schedule

3. STUDY RESULTS

3.1 MISSION AND SPACECRAFT ANALYSES

A major effort in the first month of the project was the analysis of the five specified interplanetary missions to determine realistic spacecraft configurations and mission profiles for each. Two spacecraft models were defined for each mission based on consideration of propulsion capabilities, scientific objectives, estimated power levels, and spacecraft geometry. To expedite these analyses in view of the short time available, spacecraft configurations were based on adaptation of existing vehicles and design concepts including Mariner, Voyager, Advanced Planetary Probe, and Pioneer VI. The spacecraft investigations included consideration of the use of electric propulsion systems on two of the missions to produce a relatively large power requirement.

Seven of the ten spacecraft configurations resulting from these analyses were selected by JPL for further use in the power system studies. Elimination of three of the models was based on establishing a suitable balance between the number of system analyses and the depth of each within the scope of this project. Preference was given to those model configurations where the power system requirements and design constraints were based on well established technology. As a result, the models employing more advanced concepts such as electric propulsion were eliminated.

A summary of the seven selected model spacecraft configurations is shown in Table I. In each case, salient features of the spacecraft subsystems having significant effects on the power subsystem are listed. As an example, the communications transmitter represents one of the highest single loads on interplanetary missions. Therefore, the results of tradeoffs between antenna size and orientation, transmitter output power, and resultant data rate capabilities are included. In the various missions, different types of transmitters were assumed to reflect a broader spectrum of input power characteristics. These included the travelling wave tube, Klystron, and solid-state types.

Mission Definition Spacecraft Type	1 0.3 AU Probe (or Mercury Flyby) Mariner Class With Variable-Angle Array	2 Venus Orbiter No. 1 Mariner Class With Orbit Insertion Engine
Primary mission objectives	1. Interplanetary particles and fields 2. Mercury scan	1. Interplanetary and plane tary particles and fiel 2. Venus scan
Mission C_3 (km^2/sec^2)	91 (50 to 60 for Mercury flyby)	14
Launch vehicle	Atlas/Centaur/HEKS or Titan IIIC/Centaur	Atlas/Centaur
Spacecraft injected weight (lb)	900	1500
Mission duration (yr) Transit Orbit	 {0.25 to perihelion {0.25 - 0.32 to Mercury	 0.4 0.5
Approx Power capability (w) At Earth At target (planet)	 350 350	 250 300
Weight breakdown (lb) Injected weight Propellant exp en route	 900 (4 lb midcourse, if Mercury flyby)	 1490 60
Propellant exp orbit insertion	-	750
Lander or entry capsule	-	-
Total weight expended	-	810
Total weight remaining	900	680
Science payload	60	50
Orbit characteristics Period (Earth days)	 -	 0.74, 1.52
Size (planetary radii from center of planet)	-	1.5 x 9.
Inclination	-	0 deg
Worst-case eclipse (hr)	-	2.2
Configuration	Octagonal body, roll axis toward sun. Gimbaled antenna and most experiment sensors away from sun.	Mariner II (Venus), with or insertion engine incorpora so as to point toward sun along roll axis. Thrust ≈ 400 lb.
Stabilization and control	3-axis stabilized, using sun and Canopus optical sensors for errors, and gas jets. (Mariner).	3-axis stabilized, using sun and Canopus optical senso and gas jets. Gimbaled engines and gyros during firing.
Communications (downlink to 210-ft dish)	3-ft (Mariner) dish (23.3 db), double gimbaled, and 20-w TWT transmitter gives 650 b/sec at 1.6 AU. (Earth-Spacecraft distance)	3-ft (Mariner) dish (23.3 db double gimbaled, and 10- solid-state transmitter: 3000 b/sec at 0.5 AU (Ear s/c distance at encounter) 250 b/sec at 1.7 AU (1 yea after launch)
Thermal control	Reflecting shield on sun side of equipment compartment.	Standard Mariner
Estimated solar array size and configuration	Four panels totaling 75 ft^2 extend as elements of a cross from spacecraft perpendicular to roll axis. Each panel is oriented about its axis for temperature control.	Two panels totaling 40 ft^2.

Table I. Model Spacecraft Configurations

	3 Venus Orbiter No. 2 Voyager Class With Entry Probe	4 Mars Orbiter Voyager Class (1973) Second-Generation With Lander	5 5.2 AU Probe APP Spin-Stabilized Class	6 Jupiter Orbiter No. 1 APP Class Second Generation	7 Jupiter Orbiter No. 2 Voyager Class With Multiple Entry Probes
- is	1. Venus environment 2. Venus atmosphere (scan and probe) 3. Interplanetary environment	1. Interplanetary/planetary science 2. Mars environment, atmosphere, and surface data (including biological data, if any)	1. Interplanetary particles and fields 2. Jupiter scan	1. Interplanetary exploration 2. Jupiter environment and orbital scan	1. Planetary/interplanetary data 2. Jupiter orbiter/entry probes
	14	<25	85 or 95 (Jupiter flyby)	90 to 100	90 to 100
	Saturn IB/Centaur (or two larger vehicles on one Saturn V)	Saturn V (two spacecraft per launch)	Atlas/Centaur/TE-364 ($C_3 \le 86$) or Atlas/Centaur/HEKS (crowded)	Saturn IB/Centaur/HEKS	Saturn V
	9000	20,500	650	2800	16,000
	0.4 0.5	0.5 0.5	2.0 -	2.0 0.5	2.0 0.5
	1000 1000	1010 600	>5000 200	>7000 300	>14,000 600
	9150 50	20,500 1,400	650 -	2800 80	16,000 170
	4600	9,650 plus 320 lb for orbit trim	-	1100	6,400
	1000 5650 3500	3,000 14,370 6,130	- - 650	- 1180 1620	1,000 7,570 8,430
	250	400	50	250	500
	0.74, 1.52	0.60	-	8.45	8.45
	1.5 x 9,	1.6 x 7	-	1.5 x 32 (ΔV ≈ 1.4 km/sec	1.5 x 32
	0 deg	45 deg	-	0 deg	0 deg
	2.2	2.0	-	1.6	1.6
it ted	Similar to TRW Mars Voyager (Phase IA Task B, using LEM stage), but scaled down to 2500 lb thrust, 9000 lb injected weight.	Sun/Canopus oriented. 3-axis stabilized with fixed solar array and gimbaled h.g. antenna dish. Deployed planetary scan platform. Basic spaceframe is octagonal, with liquid propellant retro stage.	Similar to APP spin-stabilized 500 lb spacecraft. Solar panels surrounding 7-ft D dish.	First sun/Canopus oriented; later Earth/Canopus oriented; large fixed antenna. Deployed solar panels.	Same as 6
's	3-axis, using sun and Canopus optical sensors and gas jets. Gimbaled engines and gyros during firing.	3-axis stabilized; requires sun and Canopus sensors, gyro package, possibly Mars sensors. TVC by retro engine gimbals. MC maneuvers by throttled retro.	Spin-stabilized. Axis near sun until 1.3 AU, then directed toward Earth. Conical scan RF tracking and jet preces	3-axis stabilized; gas jets; sun and Canopus sensors plus gyro package. Bias correction for Earth pointing based on signal strength. TVC by jet vanes.	Same as 6
), v th- r	6-ft dish (29.3 db), double-gimbaled, and 20-w TWT transmitter: 25,000 b/sec at 0.5 AU (encounter) 2,000 b/sec at 1.7 AU (1 year after launch)	12-ft paraboloid dish, gimbal mounted. 50-w TWT transmitter 15,000 b/sec at 2.6 AU - b/sec (end of mission)	7-ft dish (30.9 db), body-mounted, 20-w, Klystron transmitter. 270 b/sec at 6.0 AU.	32-ft dia paraboloid antenna 10-w TWT transmitter 2800 b/sec at 6 AU	Same as 6, except 40-w TWT 11,000 b/sec
	Louvers on equipment bays	Louvered equipment mounting panels, aluminized Mylar insulation. Thermostatically controlled heaters; thermal control of lander to be included.	Insulation from sun; thermal switches.	Insulation from sun; thermal switches or louvers	Same as 6
	Four panels totaling 140 ft^2.	20-ft dia circular array around retro engine nozzle. Eight fixed modular array plates; 280 ft^2; 290 lb.	Panels (475 ft^2, 250 lb) deployed from perimeter of 7 ft D rigid antenna and unfolded.	Deployed 8-panel array (each 10 x 10 ft) around sunflower antenna dish. Sequential deployment of solar array and antenna; (must withstand orbit insertion loads.	Same (but each panel 12.5 x 16 ft)

Mission profiles as shown in Figures 2 through 5 were prepared to show variations in earth-spacecraft and sun-spacecraft distances with mission time. Significant mission events such as midcourse maneuvers, planetary encounter, and orbit insertion were identified. In addition, the angle between the sun and the earth as viewed from the spacecraft was plotted as a function of mission time. This latter characteristic is particularly significant for the Jupiter missions where both the antenna and solar panels are earth oriented after reaching a sun-spacecraft distance of approximately 1.3 AU.

Of major interest in the power system analysis for orbital missions are the eclipse time and sunlight time for any given orbit and the variations in these parameters during the assumed 6-month orbital phase of the missions. Detailed analyses of possible orbit parameters for the Venus and Jupiter missions were necessarily beyond the scope of this study. Therefore, orbits were assumed to be in the ecliptic plane for these planets. The Mars orbit selection was based on analyses performed in the course of TRW's Voyager studies. The orbit parameters and variations in eclipse duration for the Mars and Venus missions are shown in Figures 6 and 7. The parameters for the assumed Jupiter orbit are as follows:

Orbit Period	203 hr
Eclipse Duration	1.6 hr (constant)
Periapsis Altitude	105,000 km
Apoapsis Altitude	2,170,000 km

3.2 MODEL POWER REQUIREMENTS

The model spacecraft configurations were analyzed to define typical equipment categories required in each of the subsystems (i.e., stabilization and control, communications and data handling, propulsion, thermal control, and science/payload). These equipment categories were investigated for each model and their power consumption was estimated as a

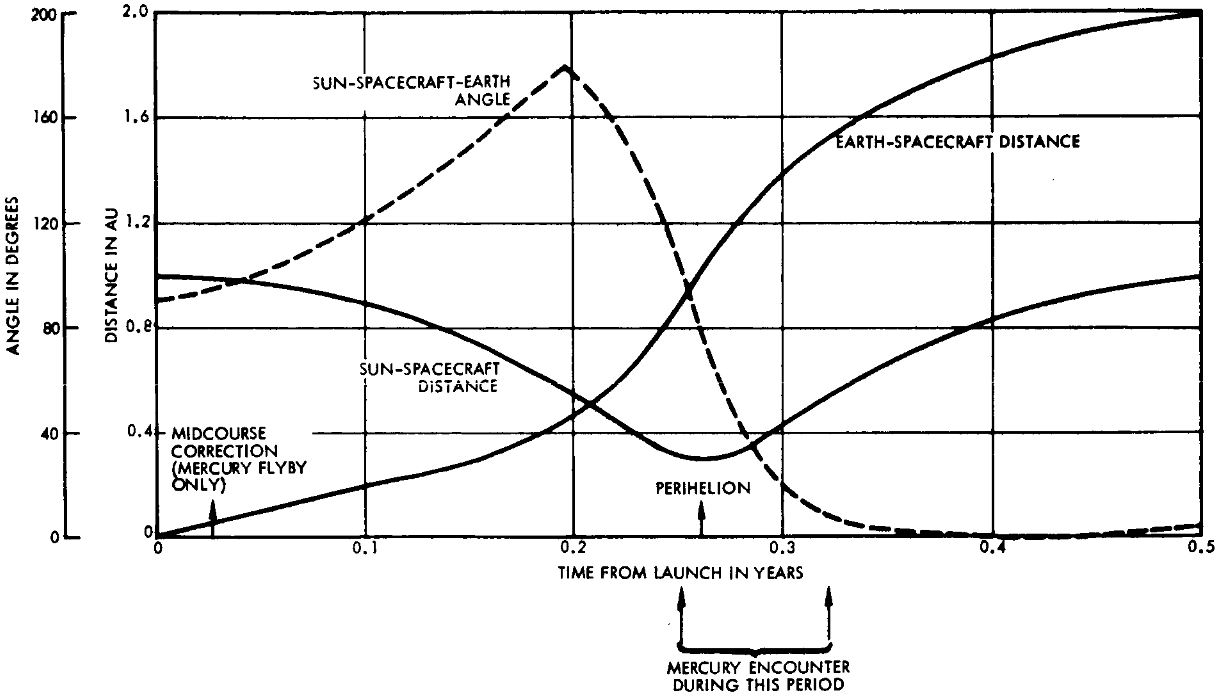

Figure 2. Mission Profile: 0.3 AU Probe, or Mercury Flyby

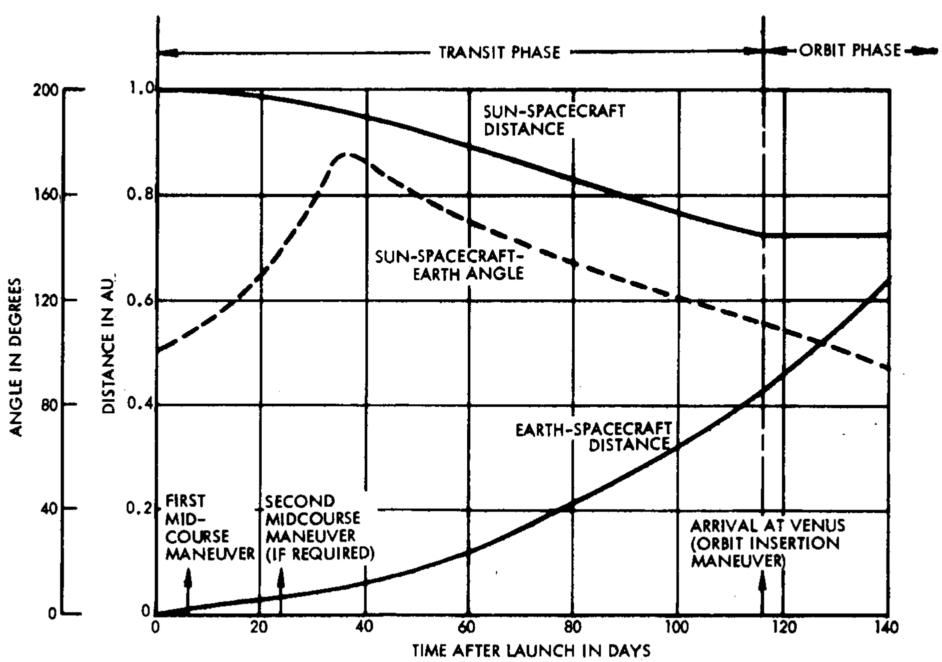

Figure 3. Mission Profile: Venus Orbiter

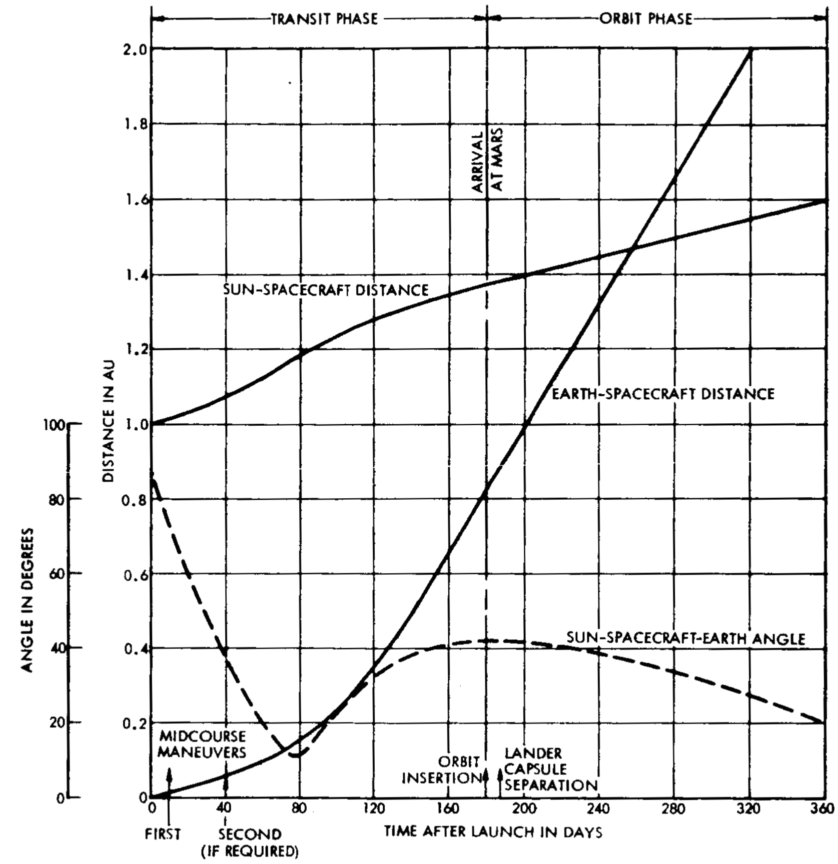

Figure 4. Mission Profile: Mars Orbiter Mission No. 2 (Launch May 1971)

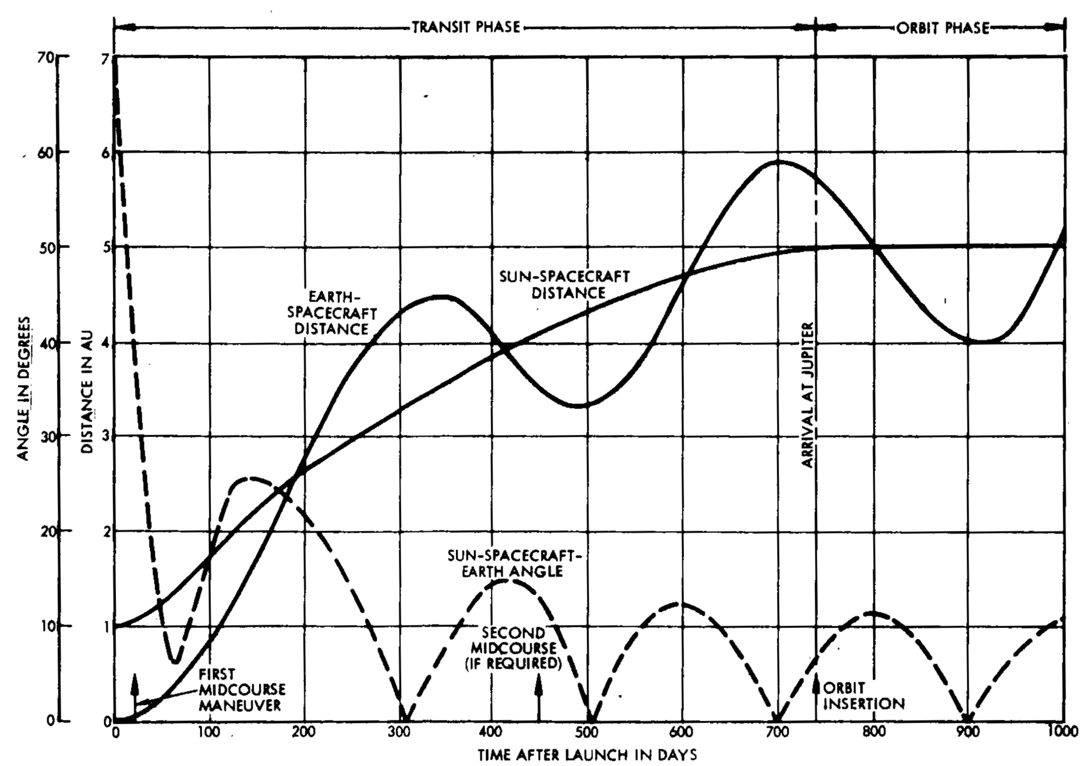

Figure 5. Mission Profile: Jupiter Orbiter Mission (Launch March 1972)

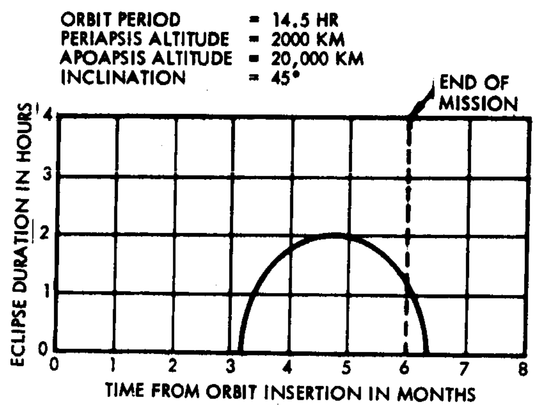

Figure 6. Eclipse Durations for Assumed Mars Orbit

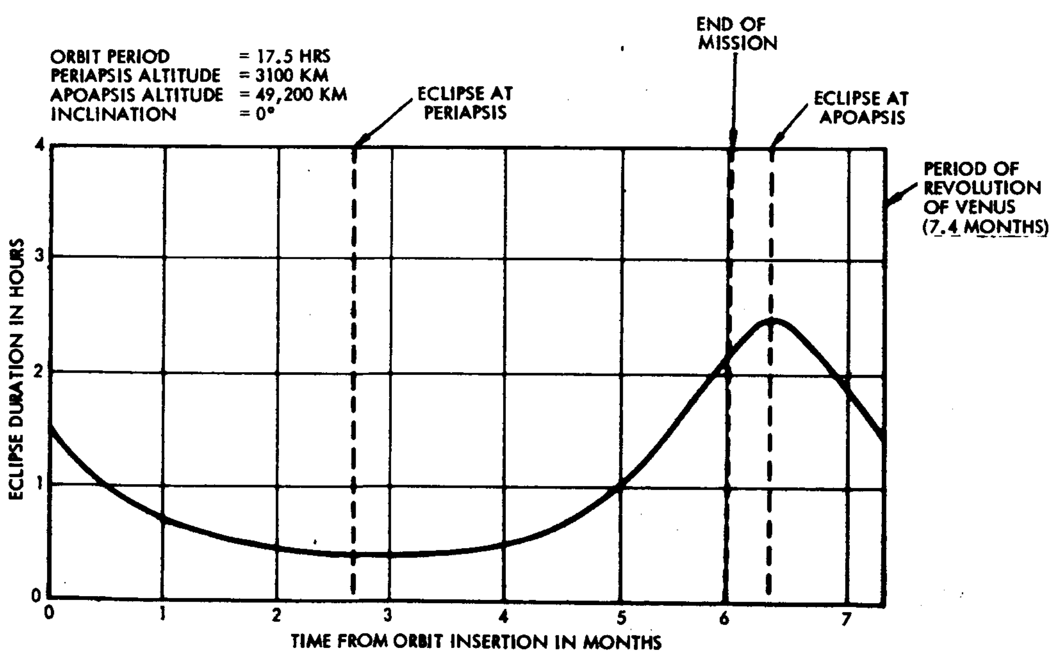

Figure 7. Eclipse Durations for Assumed Venus Orbit

function of mission phase for each case. Typical load/power requirements are shown in Tables II and III as examples of the data developed for each spacecraft model. These estimates were based primarily on load data from existing spacecraft designs such as Mariner, Pioneer, and Voyager. A significant result of these analyses was the determination that power levels in the largest spacecraft configurations fell in the lower end of the 200- to 4000-w range originally specified for analysis. The investigations of probable scientific experiments to be performed on these missions disclosed that, in most cases, individual equipment power levels of less than 10 w would adequately fulfill the scientific objectives. Television systems requiring approximately 25 w of power constituted the highest single equipment requirement in the science category. Relatively high power requirements for thermal control of lander/probe payloads were assumed for the orbiting spacecraft missions based on the 200-w requirement used in the Voyager studies. In most cases, this requirement represents the most significant single load in the spacecraft in terms of its power consumption. A second major power-consuming load is the transmitter required to achieve suitable data rates at the extreme distances being considered in these studies. Use of a 32-ft diameter paraboloid antenna at the large earth-spacecraft distances encountered in the Jupiter orbiter missions permitted selection of a relatively low-power transmitter having a 40-w output rating, and requiring an input power level of 135 w. Higher radiated power levels with a smaller antenna to yield the same 11,000 bit per sec data rate were judged to produce a less desirable overall system tradeoff between antenna weight and combined power system-transmitter weight. The largest transmitter considered in these evaluations was a 100-w TWT which was judged to represent a reasonable upper limit on state-of-the-art advancements for flight usage during the 1970 to 1980 time period assumed in the study.

The various load equipment groupings were analyzed further to ascertain their typical input voltage levels and voltage regulation requirements after power conditioning. Consideration was given to the increased use of integrated circuits in new designs for control systems and data handling equipment. This was reflected in an increase in the percentage

Table II. Conditioned Power Requirements (watts) as Function of Mission Phase (Jupiter Flyby)

LOAD GROUP	EQUIPMENT	PRE-LAUNCH	LAUNCH	ACQUIRE	CRUISE	MANEUVER	ENCOUNT	PLAYBACK
Stab. and control	Gyros and Electronics	(Not used for this model)						
	Sensors (sun)	(Negligible power consumption)						
	Control Elect.	5	5	5	5	5	5	5
	Valves	0	0	0/0/6*	0/0/6	0/0/6	0/0/6	0/0/6
Propulsion	Valves	0	0	0	0	Squibs	0	0
	Actuators	(Not used for this model)						
Integration	Comp. Sequencers	5	5	5	5	5	5	5
	Deploy. Actuators		Squibs					
Thermal Control	Heaters	0	0	0	0/10/50	0	50	50
Communications	Transmitter **	1	1	1/80/80	80	80	80	80
	Tape Recorder	0	0	0	0	0	6	4
	Data Handling	10	10	10	10	10	10	10
	Cmd Receiver	2	2	2	2	2	2	2
	Cmd Decoder	2	2	2	2	2	2	2
	Relay Receiver	(Not used for this model)						
	Antenna Orient.	(Not used for this model)						
Science/Payload	Lander/Probe	(Not used for this model)						
	TV System	0	0	0	0	0	17	0
	Exp. Pkg. Orient.	0	0	0	0	0	7	0
	Magnetometer	0	0	0	4	4	4	4
	Trap. Radiation Det	0	0	0	2	2	2	2
	Plasma Probe	0	0	0	2	2	2	2
	Cosmic Dust	0	0	0	1	1	1	1
	Dual Freq. Rec	0	0	0	2	2	2	2
	Radiometer, IR	0	0	0	7	7	7	7
	Cosmic Ray (2)	0	0	0	4	4	4	4
Total Conditioned Power		25	25	104	136	136	206	180

*Indicates Min/Ave/Max power levels.

**20 w klystron and driver

Table III. Conditioned Power Requirements (watts) as Function of Mission Phase (Jupiter Orbiter No. 2)

LOAD GROUP	EQUIPMENT	PRE-LAUNCH	LAUNCH	ACQUIRE	CRUISE	MIDCOURSE MANEUVER	ORBIT INSERTION	ORBIT (SUN)	ORBIT (ECLIPSE)	SEPARATE PROBE	TRACK PROBES	ORBIT (SUN)	ORBIT (ECLIPSE)	NOTES
Stab. and Control	Gyros and Electronics	25	25	2	0	25	25	0	25	0	0	0	25	
	Sensors	4	4	4	4	4	4	4	4	4	4	4	4	
	Control Elect.	10	10	10/10/50	10/10/50	10/10/50	10/10/50	10/10/50	10/10/50	10/10/50	10/10/50	10/10/50	10/10/50	* Min/Ave/Max power levels
	Valves	0	0	0/0/50	0/0/50	0/1/50	0/1/50	0/0/50	0/0/50	0/0/50	0/0/50	0/0/50	0/0/50	
Propulsion	Valves	0	0	0	0	0/4/*	0/8/*	0	0	0	0	0	0	*25W Solenoids + Squibs
	Actuators	0	0	0	0	0/8/15	0/8/15	0	0	0	0	0	0	
Integration	Comp Sequencer	20	20	20	20	20	20	20	20	20	20	20	20	
	Deploy. Actuators		Squibs							Squibs				
Thermal Control	Heaters	0	0	0	200	200	200	200	250	200	200	200	250	
Communications	Transmitter	5	5	5/135/135	135	135	135	135	135	135	135	135	135	40W TWT + Driver
	Tape Recorder	0	0	0	0/4/12	0/4/12	0/4/12	0/17/60	0/17/60	0/17/60	0/17/60	0/17/60	0/17/60	
	Data Handling	10	10	40	40	40	40	40	40	40	40	40	40	
	Cmd Receiver	8	8	8	8	8	8	8	8	8	8	8	8	
	Cmd Decoder	5	5	5	5	5	5	5	5	5	5	5	5	
	Relay Receiver	0	0	0	0	0	0	0	0	2	2	2	2	
	Antenna Orient	(Not used for this model)												
Science/ Payload	Lander/Probe	0	0	0/150/150	150	150	150	150	150	150	0*	0*	0*	*Ave level decreases to zero with separation of last probe
	TV System	0	0	0	0	0	0	26	26	26	26	26	26	
	Exp Pkg Orient	0	0	0	0	0	0	2/4/40	2/4/40	2/4/40	2/4/40	2/4/40	2/4/40	
	Cosmic Ray (2)	0	0	0	10	10	10	10	10	10	10	10	10	
	Plasma Probe	0	0	0	5	5	5	5	5	5	5	5	5	
	Magnetometers (2)	0	0	0	6	6	9	3	3	3	3	3	3	
	Micrometeoroid	0	0	0	2	2	2	2	2	2	2	2	2	
	Radio Propagation	0	0	0	3	3	3	3	3	3	3	3	3	
	Trap. Radiation	0	0	0	5	5	5	5	5	5	5	5	5	
	Radiometers (2)	0	0	0	6	0	0	20	20	20	20	20	20	
	Auroral Detector	0	0	0	0	0	0	2	2	2	2	2	2	
	Spectrometer	0	0	0	0	0	0	15	15	15	15	15	15	
	Topside Sounder	0	0	0	0	0	0	3	3	3	3	3	3	
Total Conditioned Power (AVE)		87	87	397	611	645	657	685	762	689	539/689	532	614	

of total input power utilized at the lower voltage levels in comparison to that of existing equipment. For each equipment category, the input power was apportioned among the required input voltages. These data, together with the load requirements data, therefore, define the required outputs of the power subsystem for each model spacecraft.

3.3 SOLAR ARRAY ANALYSIS

Representative solar array output current-voltage characteristics were computed for each mission as functions of sun-spacecraft distance. The solar cells used in the analysis of inbound missions to Venus and Mercury were those of a specially designed 1 x 2 cm size having a base resistivity of 10 ohm-cm, 10 percent AMO efficiency, and cover slides with a 420μ cutoff filter. These cells were fabricated for high light intensity operation with a very low value of series resistance (approximately 0.2 ohms) through use of twelve grids rather than the usual five. The solar cell characteristics used in the analysis of the outbound missions to Mars and Jupiter were those of a 2 x 2 cm, 10.5 percent efficiency, 10 ohm-cm type covered by a 420μ cutoff filter.

Output calculations in each case were based on a 10 series by 10 parallel solar cell array and utilized TRW Computer Programs AM 118 and AM 142. The first of these programs is designed for the missions with decreasing solar intensity and the second program takes into account the effects of high solar intensity on cell performance as encountered on the Mercury and Venus models. In these analyses, a solar flare radiation environment equivalent to 10^{14} 1-mev electrons per cm^2 per year near the Earth (1 AU) was assumed. It was further assumed that the radiation levels at other than 1 AU varied inversely with the square of the sun-spacecraft distance.

Representative results of these calculations are shown in Figures 8 and 9 for the Mercury and Mars missions, respectively. In addition to the array current-voltage characteristics at selected points in the mission, the variation in solar array current and voltage corresponding to the maximum power point throughout the mission is also indicated. For the Mercury mission, the maximum array power is shown to increase

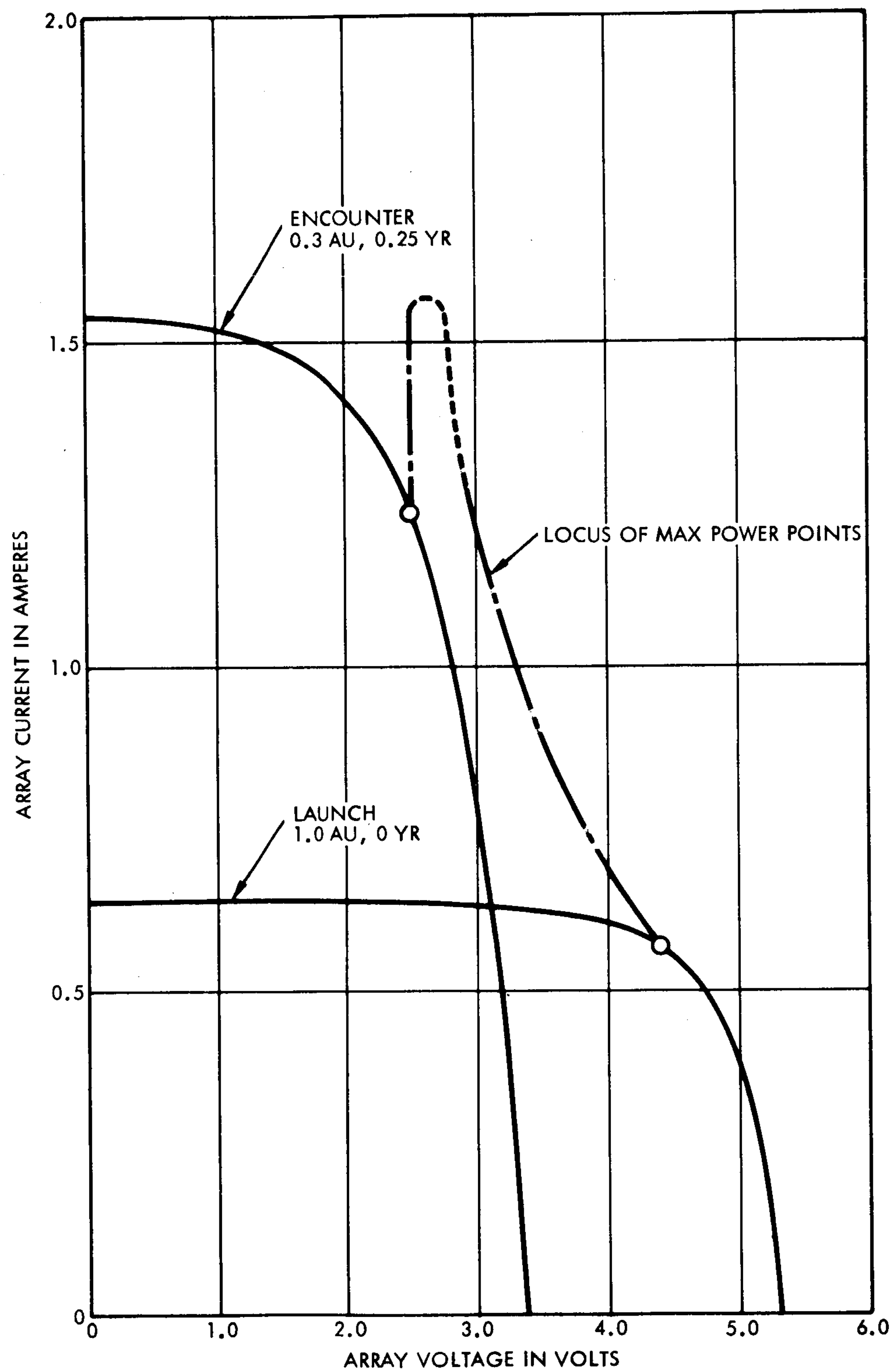

Figure 8. Mercury Flyby Solar Array Characteristics
10 Series x 10 Parallel Cells
(1 x 2 cm cells)

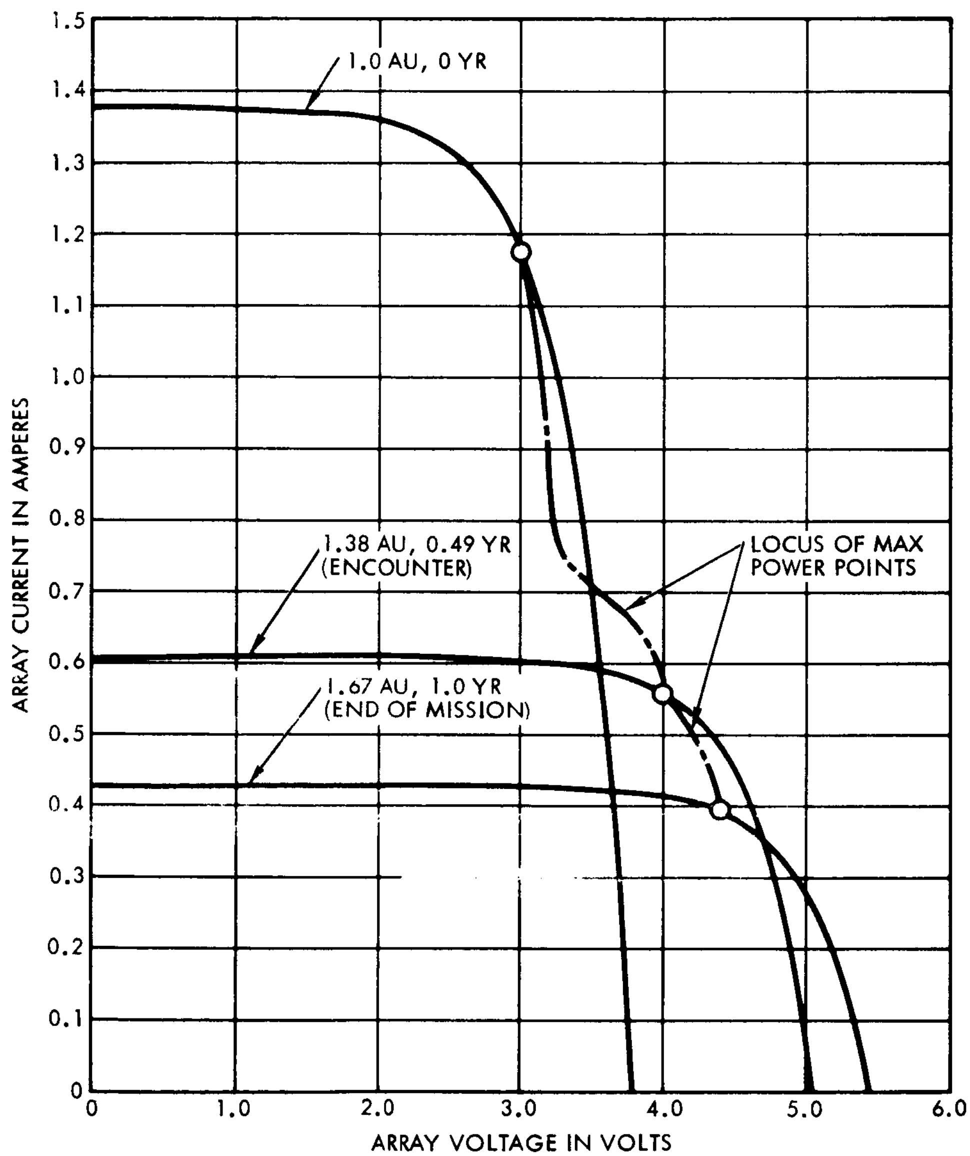

Figure 9. Mars Orbiter Solar Array Characteristics
10 Series x 10 Parallel Cells
(2 x 2 cm cells)

to a maximum and then decrease at lower values of sun-spacecraft distance. This results from tilting the solar panels from their sun-oriented position to prevent excessive cell temperatures at the lower values of sun-spacecraft distance.

3.4 BASELINE POWER SYSTEM ANALYSIS

Analysis of alternative baseline power system configurations for each model set of requirements was initiated during the second month of the project. Initially, comparisons were made of the solar array maximum power capability and the total conditioned load power requirement as functions of mission time to define apparent design points for each model. These design points represent the condition during each mission of minimum solar array capability relative to the required load. For the orbiting missions, the required load included estimated battery charging power requirements. It was determined that the design points for all missions occurred either at minimum or maximum sun-spacecraft distance (AU). Alternative system configurations were compared with respect to their ability to make maximum use of the solar array power capability at these design points and to provide positive power margins at all other times in any given mission.

Investigations of candidate power system configurations were based on progression from generalized system concepts to specific baseline implementations as shown in the flow diagram, Figure 10. Initially, power systems were divided into two generalized concepts as shown in Figure 11. From these two concepts, the basic functional power system configurations shown in Figure 12 were developed. Referring to Figure 11, the first generalized concept combines the battery and solar array outputs at an unregulated bus with suitable controls. The unregulated bus supplies line regulation and power conditioning equipment which, in turn, supplies the regulated outputs of the system. In addition, the unregulated bus can directly supply certain of the spacecraft loads such as heaters and solenoids. The second approach employs regulators for both the solar array and battery to permit their electrical connection to a regulated dc bus which supplies the power conditioning equipment and direct connected loads.

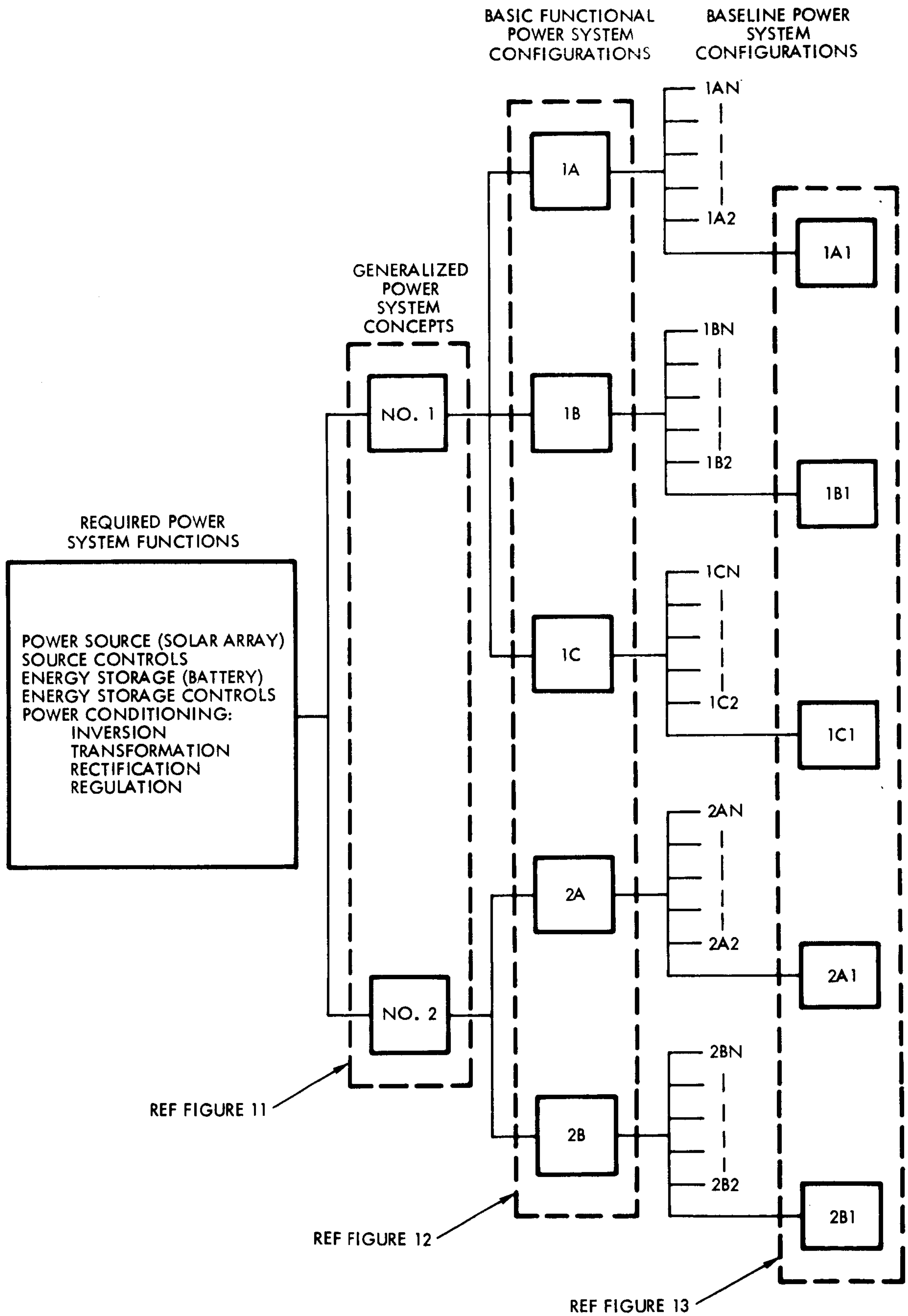

Figure 10. Flow Diagram – Baseline System Analysis

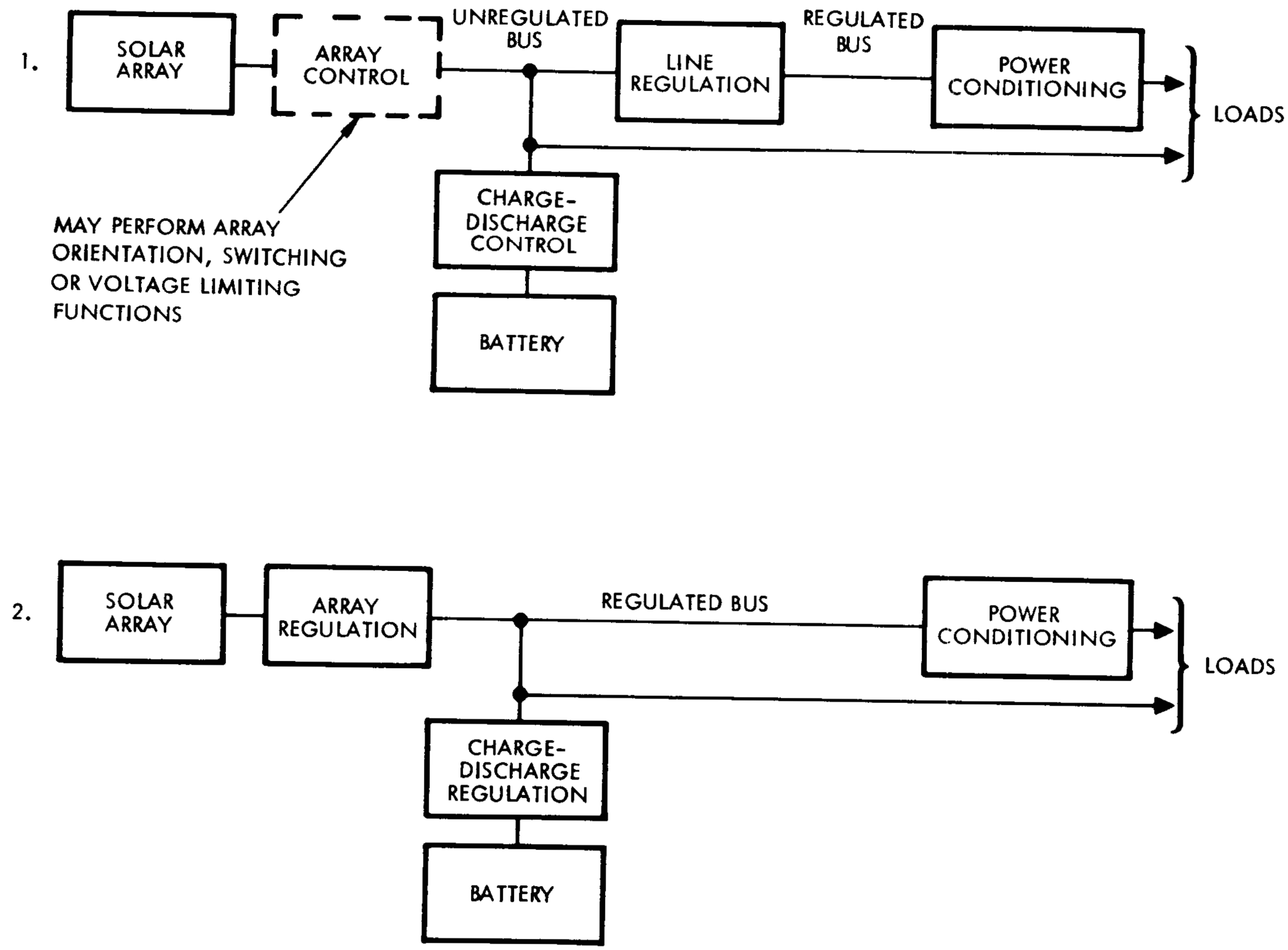

Figure 11. Generalized Power System Concepts

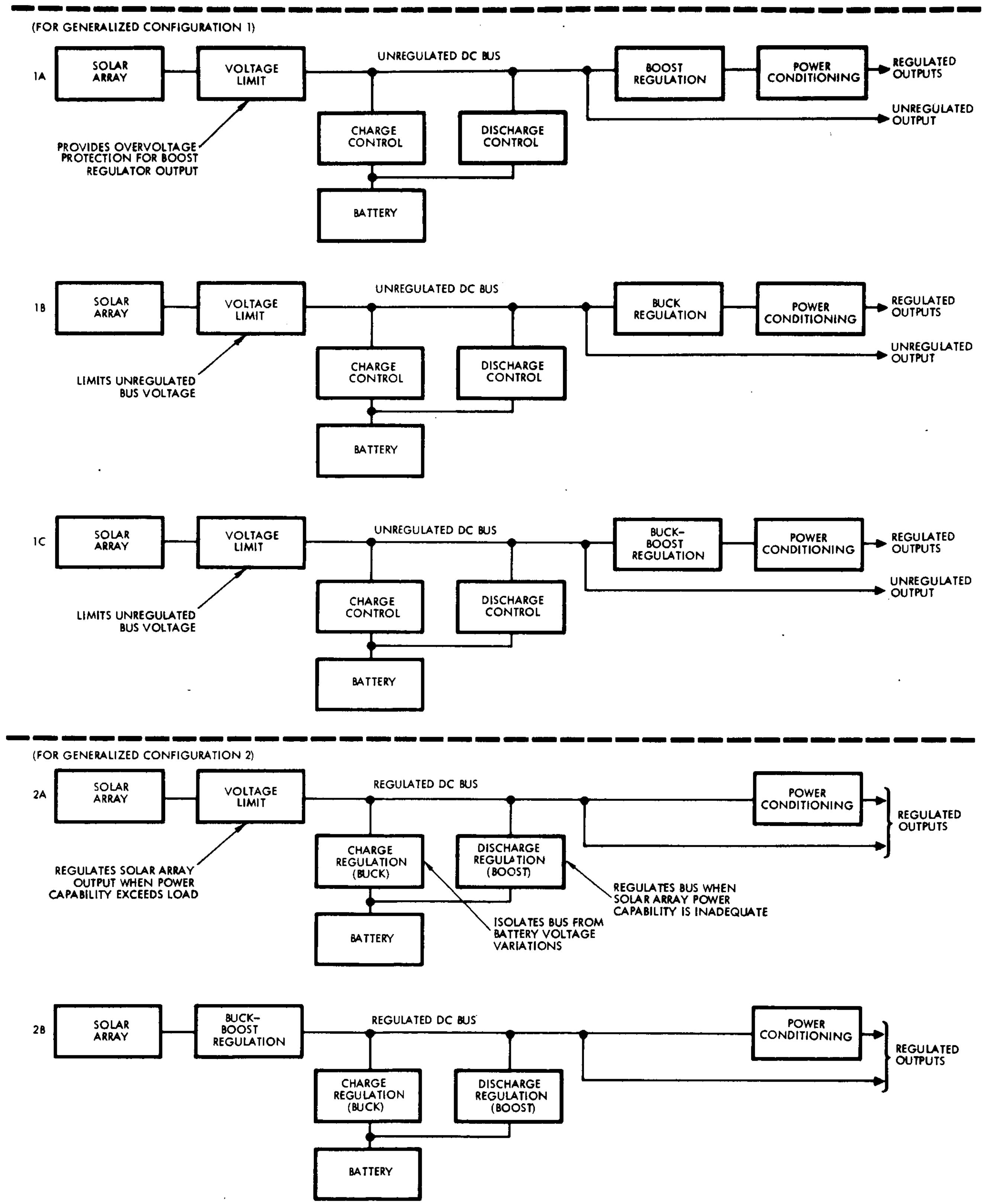

Figure 12. Functional Power System Configurations

From these five functional system approaches, baseline system configurations were determined, based on several specific designs for each functional element of each basic configuration. Figure 13 shows five examples of baseline systems, one for each basic system configuration.

The basic functional configurations of Figure 12 were selected on the basis of their compatibility with the variations in load and solar array characteristics encountered during the interplanetary missions under consideration. In each system configuration, specific functions are identified which satisfy the regulation requirements of the applicable generalized concept. For generalized configuration 1, the three alternative approaches to accomplishing the line regulation fun ction are shown. In general, voltage boosting, (configuration 1A) tends to minimize regulation losses at maximum sun-spacecraft distance (AU). Conversely, bucking line regulation (configuration 1B) tends to minimize series losses at minimum AU. The combined buck-boost approach (configuration 1C) can be optimized with respect to efficiency at any selected value of AU. Voltage limiting of the array output is essential in configuration 1A to prevent overvoltage conditions at the regulated input to the power conditioning equipment. In configurations 1B and 1C, voltage limiting of the array is required only if the loads connected to the unregulated bus cannot tolerate maximum solar array voltage levels. The two alternative approaches to providing the solar array regulation function for generalized configuration 2 also are shown. The voltage limiting approach of configuration 2A requires that the regulated bus voltage be selected at or below the minimum steady-state voltage of the array. This approach, therefore, is similar to configuration 1B in that it tends to minimize system losses at minimum AU. The use of a buck-boost array regulation approach in configuration 2B is similar to that of configuration 1C in that it permits efficiency optimization at any AU value. It should be noted that the functions shown, in many cases, could be implemented in several different ways. For example, the array voltage limiter function could use either series or shunt regulator circuits and each of these, in turn, could be implemented using either dissipative or switching (pulse width modulation) techniques.

Preliminary indications of optimized baseline system configurations were arrived at by implementing the control functions of each system from Figure 12 in a manner which provided for maximum usage of the solar array power capability at the design point. The power conditioning functions of inverting, transforming, rectifying, filtering, and regulating are common to all configurations and, therefore, were excluded from these analyses. A figure of merit (Z) was developed relating power available at the regulated bus to maximum solar array power for the design point conditions as follows:

$$Z = \frac{P_{avail}}{P_{sa_{max}}} = \frac{HP_{sa_v}}{P_{sa_{max}}}$$

where

H = system efficiency = $\dfrac{1 + t_e/t_d}{\frac{1}{\eta_r}\,(1 + t_e/t_d\,\eta_s)}$

P_{sa_v} = solar array output power at minimum operating voltage

$P_{sa_{max}}$ = solar array output power at voltage corresponding to maximum power point

t_e/t_d = ratio of eclipse time to sunlight time per orbit

η_r = efficiency of solar array regulator and/or line regulators

η_s = product of battery storage efficiency, charge control efficiency, and discharge control efficiency

Examples of baseline configurations which resulted from these figure-of-merit determinations are illustrated in Figure 13. These specific examples are optimized for maximum efficiency at maximum AU. It should be noted that these analyses have indicated approaches which minimize solar array size and weight by maximizing the figure of merit (Z) at the design point. In each case, it was necessary to verify that changes in the figure of merit during the mission did not produce a new point as a result of decreased system efficiency at conditions other than the original design point.

Additional analysis of baseline systems is in progress to determine the reliability and weight of all the configurations considered and to assess the advantages and disadvantages of the various configurations relative to electromagnetic compatibility, thermal interfaces with the spacecraft, and flexibility with respect to load growth. The configurations under consideration include those which provide for maximum solar array power utilization as well as simpler, less efficient versions of the same basic functional configurations. Efforts to date have consisted principally of determining electronic parts count, efficiency, and weight data for each component of the various baseline system configurations. Comparative analyses will be completed for each of these systems during the second quarter of the program to provide quantitative reliability and weight tradeoffs and to serve as a basis for the ensuing analysis of methods for improving system reliability.

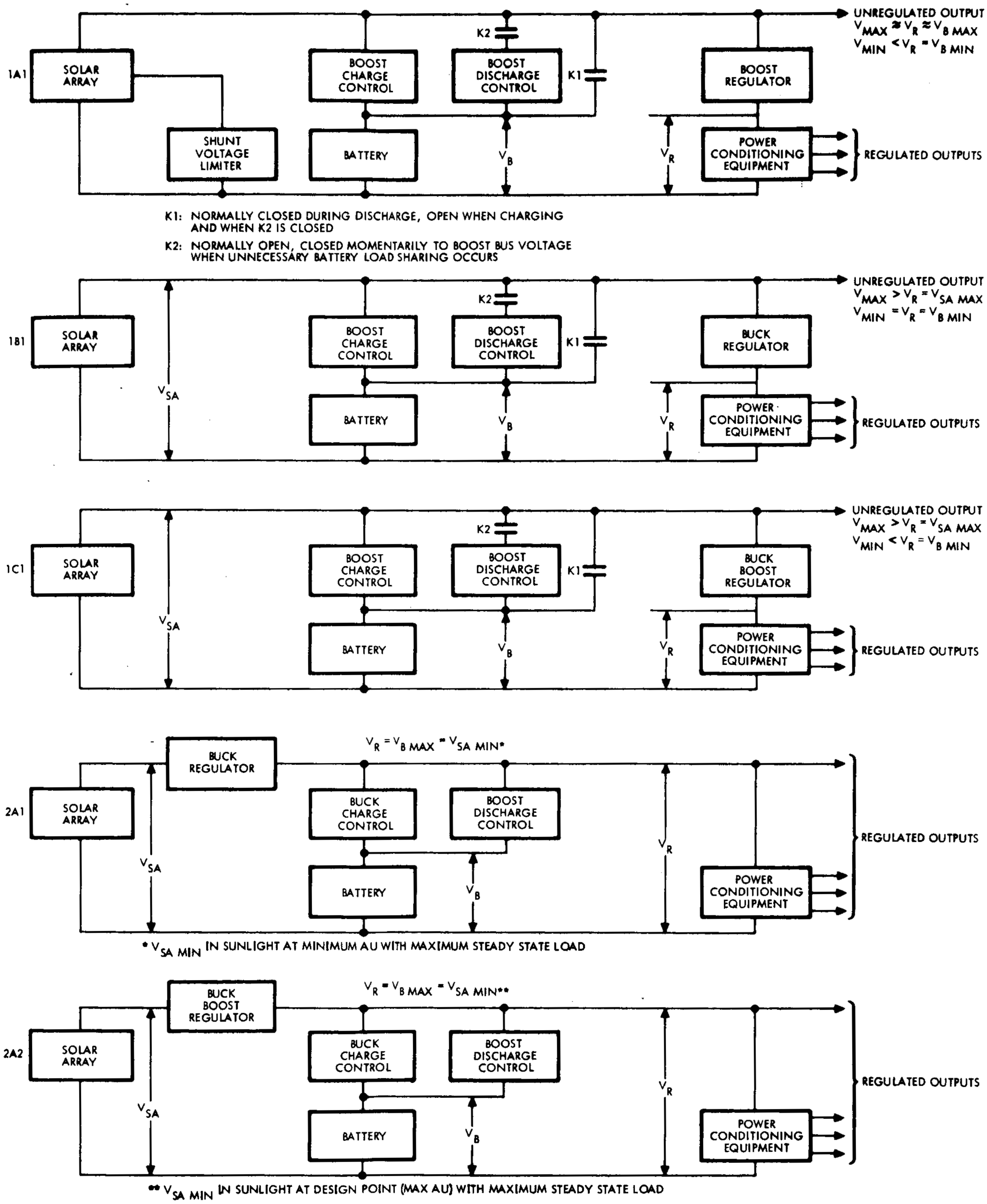

Figure 13. Baseline Power System Configurations for Maximum Power Utilization at Maximum AU